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PERFORMANCE OF SWIRL-CAN TURBOJET COMBUSTORS AT SIMULATED SUPERSONIC COMBUSTOR-INLET CONDITIONS

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ABSTRACT

A turbojet combustor comprising an array of swirl cans was evaluated in a rectangular duct at conditions simulating supersonic flight. Tests were conducted at a pressure of 3 atmospheres, inlet-air temperatures of 540° and 1140° F (556 and 889 K), and combustor reference velocities up to 190 feet per second (57.9 m/sec). At combustor-outlet temperatures near 2200° F (1478 K) combustion efficiencies were near 100 percent. At a diffuser-inlet Mach number of 0.3 and an outlet- to inlet-temperature ratio of 2.5, the overall pressure loss was 4.6 percent. At an inlet-air temperature of 1140° F (889 K), temperature rise variation ratios ΔTVR below 1.25 were obtained.

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SUMMARY

Swirl-can combustor elements consisting of a fuel-air mixing section, a swirler, and a diverging flameholder were tested in an array of three rows of seven each. The swirl cans were mounted in a 15-inch- (0.381-m-) long, 33°-included-angle diffuser to give a burning length of 20 inches (0.508 m) and a diffuser-inlet to exhaust-nozzle length of 39 inches (0.991 m). Tests were conducted over a range of fuel-air ratios at a pressure of 3 atmospheres, combustor-inlet temperatures of 540° and 1140° F (556 and 889 K) and combustor reference velocities up to 190 feet per second (57.9 m/sec).

In the range of fuel-air ratios required to give average combustor-outlet temperatures of 2200° F (1478 K), combustion efficiencies varied between 95 and 100 percent. At a diffuser-inlet Mach number of 0.3 and a combustor-outlet- to inlet-temperature ratio of 2.5, the overall pressure loss $\Delta P/P$ was 4.6 percent. Combustor-outlet temperature distribution improved appreciably with increasing inlet-air temperature. At a combustor reference velocity of 150 feet per second (45.7 m/sec), temperature distribution parameters ∂_{stator} and ∂_{rotor} decreased from 0.318 and 0.115, respectively, at an inlet temperature of 540° F (556 K) to 0.211 and 0.062, respectively, at an inlet-air temperature of 1140° F (889 K). Altitude blowout and reignition tests conducted at a combustor reference Mach number of 0.1 and a fuel-air ratio of 0.01 showed that at a pressure of 1.16 atmospheres no blowout occurred at inlet temperatures as low as 100° F (311 K). At a pressure of 0.75 atmosphere, blowout occurred at an inlet-air temperature at 250° F (394 K). For reignition at this condition, it was necessary to increase the inlet-air temperature to 450° F (505 K).

INTRODUCTION

Today's high-speed aircraft missions, as well as those of the future, require turbojet engines with high heat-release rates per unit volume. The Lewis Research

Center is engaged in research directed toward the solution of related jet engine combustor problems. A combustor made up of a 21-unit array of swirl-can elements located in high-velocity airstreams was investigated.

Future combustors must be short and capable of sustained performance at inlet-air temperatures above 1000° F (811 K) and at combustor-outlet temperatures above 2000° F (1366 K). In addition, combustor pressure loss must be low. These requirements for short combustor length and low pressure drop impose severe mixing problems, with the result that the combustor-outlet temperature distribution may be marginal. The high combustor-inlet and outlet temperatures coupled with high combustor-inlet pressures present severe durability problems. In addition, carbon and smoke formation may be increased (refs. 1 and 2). Combustion efficiency, on the other hand, should not present a serious problem. Although no comprehensive document on the state of the art is available at the present time, some idea of the progress made along these lines may be obtained from references 3 and 4.

This investigation was conducted to evaluate the concept of multiple combustor elements with liquid fuel at combustor-inlet conditions similar to those encountered in supersonic flight. Swirl-can combustors have been found to perform effectively with gaseous fuels (ref. 5), and with vaporized liquid fuels (ref. 6). They offer the following advantages over conventional designs:

(1) No complex fuel nozzles are required for atomization.

(2) Fuel-control orifices can be removed from the combustion zone, thus minimizing problems resulting from fuel cracking.

(3) The modular construction provides a means of adjusting combustor-outlet temperature profiles through swirl-can arrangement and through control of fuel flow to the individual swirl cans.

(4) The absence of a secondary mixing liner should lower pressure-drop requirements and reduce durability problems.

(5) The modular construction makes it possible to conduct a large portion of the developmental research in small test facilities.

The swirl-can elements consist essentially of an air-fuel mixing section followed by a diverging section which acts as the flame seat. Liquid fuel is mixed in an annulus with a small amount of combustion air and enters the conical section through a swirler. Additional combustion air enters the cone through an orifice. Most of the air flows axially past the swirl cans; some of this air recirculates in their wakes and completes the combustion reaction. Combustor walls do not extend into the secondary combustion zone. Mixing of dilution air and combustion products is facilitated by the large interface area between the hot and cold streams.

Tests were conducted in a connected-duct facility with a rectangular test section housing three rows of seven 2.9-inch- (0.0736-m-) diameter swirl cans each. Combustion efficiency, pressure loss, and combustor-outlet temperature distribution data were obtained with ASTM A-1 fuel. Tests were made over a range of fuel-air ratios at the combustor-inlet conditions shown in table I (p. 11).

TEST INSTALLATION

A 12- by 30-inch (0.305- by 0.762-m) test section housing three rows of seven swirl cans each was installed in a closed-duct test facility (fig. 1) connected to the laboratory air supply and exhaust systems. Combustion air at pressures up to 10 atmospheres was passed through an indirect-fired heat exchanger which was capable of heating the air to 600° F (589 K). For those conditions requiring a combustor-inlet temperature of 1140° F (889 K), the air was preheated further by a direct-fired (vitiating) preheater consisting of ten J71 single combustor cans. A set of baffles was installed downstream of the J71 cans to ensure a uniform temperature profile at the combustor inlet. Airflow rates and combustor pressures were regulated by remotely controlled valves upstream and downstream of the test section.

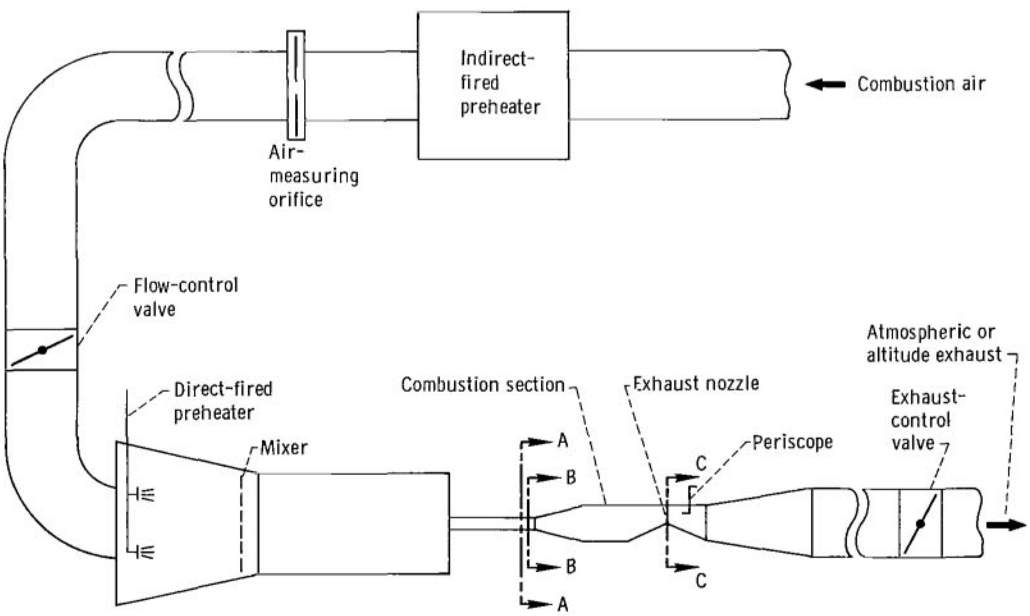

Figure 1. - Combustor installation and auxiliary equipment.

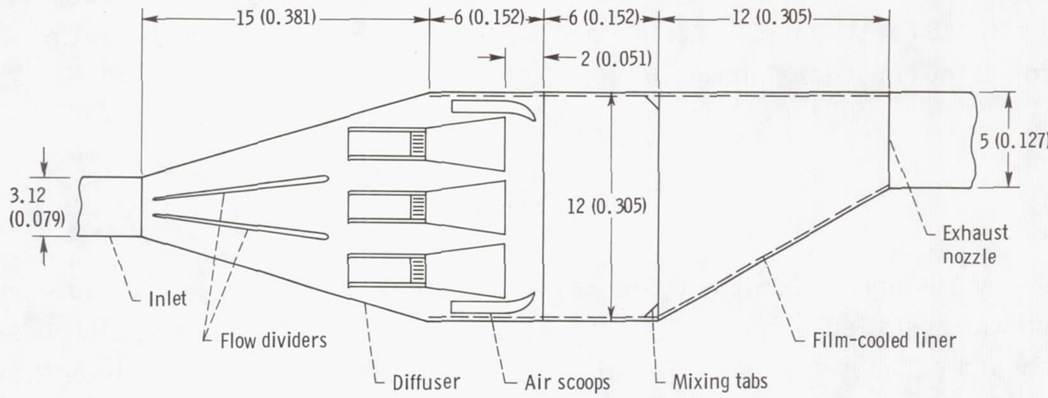

Figure 2. - Combustor installation in test section. (Dimensions are in inches (m).)

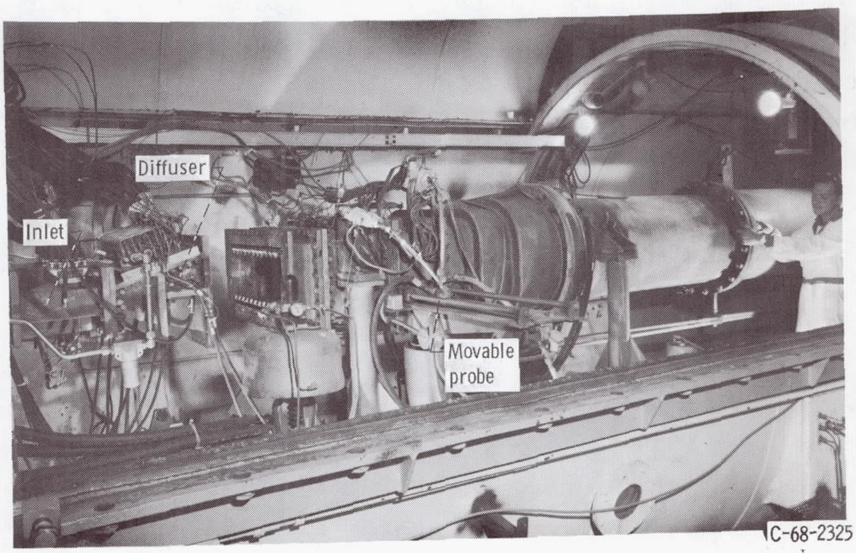

Figure 3. - Test installation.

The test sections (fig. 2), including inlet, diffuser, combustor, and exhaust nozzle, were scaled to simulate a 90° sector of a full annulus of a present-day turbojet engine with a 57-inch- (1.45-m-) diameter outer casing. For ease of fabrication, the test sections were made rectangular in cross section with a height of 12 inches (0.305 m). The diffuser had a 33° included angle and was 15 inches (0.381 m) long. Because of the steep diffuser angle, it was necessary to install a pair of flow dividers to provide a uniform velocity profile and to prevent flow separation at the walls. The inlet sections of the swirl cans were located in the diffuser just downstream of the flow dividers. The combustor length could be varied in 6-inch (0.152-m) increments by the insertion of additional test sections. Most of the tests were made with a can-outlet to exhaust-nozzle length of 20 inches (0.508 m) and an overall length (diffuser inlet to exhaust nozzle) of 39 inches (0.991 m). A film-cooled liner extending from the downstream end of the swirl cans to the exhaust nozzle was used to protect the outer housing. The test installation is shown in figure 3.

Instrumentation

Airflow rates were measured by square-edged orifices installed according to ASME specifications. Fuel flows were measured by turbine-type flowmeters, the output of which was connected to frequency-to-voltage converters.

The location of the pertinent instrumentation planes is shown in figure 1; the arrangement of the pressure and temperature probes is shown in figure 4. Pressures in the inlet section were measured by means of five rakes, each consisting of five-point total-pressure tubes, and by four wall static-pressure taps (section A-A, fig. 4). Temperatures were measured by 10 Chromel-Alumel thermocouples (section BB, fig. 4). Combustor-outlet total pressures and temperatures were recorded by means of a movable seven-point total-pressure and seven-point total-temperature rake (section C-C, fig. 4). The temperature probes were constructed of platinum - 13-percent-rhodium platinum and were of the high-recovery aspirating type (type 6, ref. 7). The average reading of four static-pressure taps located as shown in figure 4 was used as a measure of the static pressure at the exhaust nozzle. The exhaust rake is shown in figure 5.

Steady-state pressures were measured and recorded by the laboratory's Digital Automatic Multiple Pressure Recorder (DAMPR) while non-steady-state pressures were measured by strain-gage-type pressure transducers and were processed by the laboratory's Central Automatic Data Processing System (ref. 8). The thermocouple and fuel flowmeter outputs were processed by the same system. Temperature and pressure surveys at the combustor exit were made by moving the probe horizontally across the exhaust nozzle at a speed which produced approximately one reading every 1/2 inch (0.0127 m). Data needed to monitor the operation of the combustor, such as fuel flow and airflow, combustor pressures, and inlet and outlet temperatures, were also displayed in the control room. A periscope mounted downstream of the exhaust nozzle provided a view of the individual swirl-can combustors.

Calculations

Combustion efficiency was defined as the ratio of actual temperature rise to theoretical temperature rise. The amount of oxygen depletion resulting from vitiation of the combustion air was not considered significant and thus was ignored in the combustion

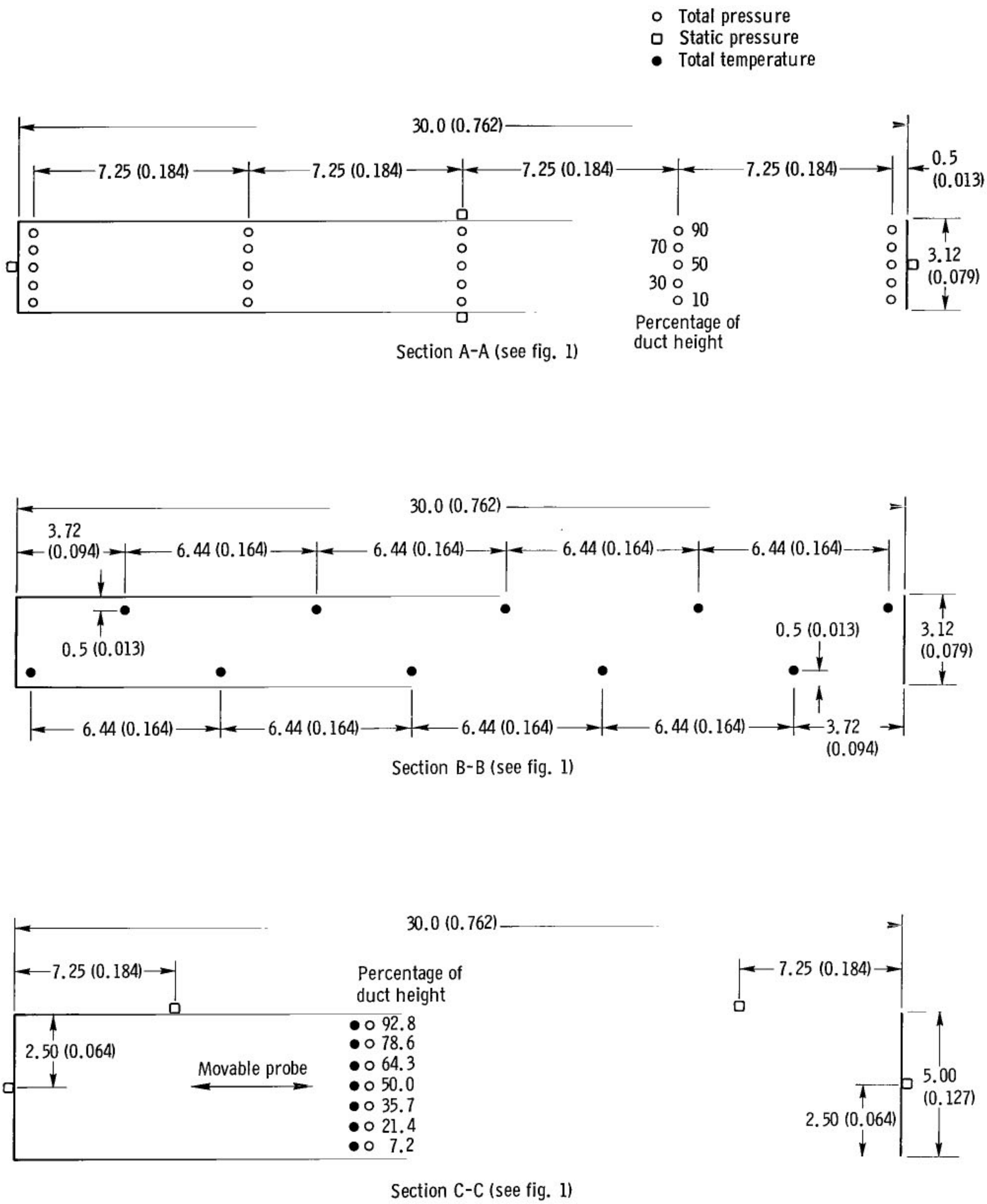

Figure 4. - Location of temperature and pressure probes in instrumentation planes. (All dimensions are in inches (m).)

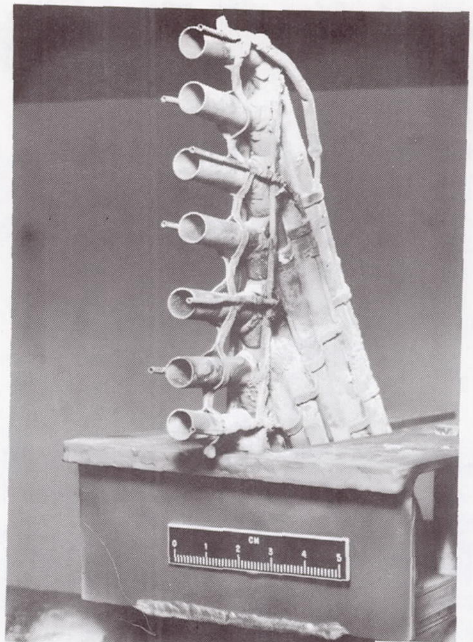

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Figure 5. - Exhaust rake.

efficiency calculations. Combustor pressure loss $\Delta P/P$ was defined by the following expression:

$$\frac{\Delta P}{P} = \frac{\text{Average inlet total pressure - Average exhaust total pressure}}{\text{Average inlet total pressure}}$$

Thus, the pressure loss includes the diffuser pressure drop.

To describe the quality of the combustor-outlet temperature profile, the following temperature distribution parameters were established:

$$\delta_{stator} = \frac{\left(T_{R, local} - T_{R, ideal}\right)_{max}}{\Delta T_{av}}$$

where $\left(T_{R, local} - T_{R, ideal}\right)_{max}$ is the largest temperature difference between the highest local temperature on any radius and the ideal temperature for that same radius, and T_{av} is the average temperature rise across the combustor.

$$\delta_{rotor} = \frac{\left(T_{R, av} - T_{R, ideal}\right)_{max}}{\Delta T_{av}}$$

where $\left(T_{R, av} - T_{R, ideal}\right)_{max}$ is the largest temperature difference between the average circumferential temperature on any radius and the ideal temperature for that same radius. The terms radial and circumferential are used as though the test section were a sector of an annulus. The ideal radial temperature profile for simulated sea-level takeoff, as well as that for cruise conditions, is typical of those encountered in advanced supersonic engines (ref. 3). The shape of the radial profile is generally dictated by the requirements of the turbine stator and rotor. In addition to the factors δ_{stator} and δ_{rotor} another parameter, used in the aircraft industry and based only on maximum and average temperature rise, was employed. It is defined as follows:

$$\Delta TVR = \frac{\text{Maximum local combustor-outlet temperature - Average combustor-inlet temperature}}{\text{Average combustor-outlet temperature - Average combustor-inlet temperature}}$$

For the combustion efficiency calculations the combustor-outlet temperatures were mass-weighted, and the average was based on the total number of readings taken in the survey. For the temperature profile calculations, the actual nonweighted temperatures were used; approximately 10 percent of the readings at each side were disregarded to eliminate sidewall effects which would not be present in a complete annular combustor.

COMBUSTORS

The swirl-can combustors used in this investigation consisted essentially of three parts: a fuel-air mixing section, a swirler, and a diverging section serving as a flame seat. A sketch of a typical swirl can is shown in figure 6. Fuel was supplied to each can through two open-end tubes with the flow-control orifices located outside the combustion chamber. The swirl cans were arranged in three rows of seven cans each with the cans staggered with respect to each other. The arrangement of the cans in the test section and the location of the fuel tubes are depicted in figure 7. The three half-cans shown in the figure were nonburning and were installed solely to provide uniform blockage.

Two sizes of swirl cans were used; one with a maximum cone diameter of 2.9 inches (0.0736 m) and one with a diameter of 3.7 inches (0.0940 m). Two types of swirlers were tested (fig. 6): a radial swirler which injected the fuel-air mixture radially inward and an axial swirler which injected the mixture in an axial direction. The open areas of the two swirlers were approximately equal. A 3/4-inch- (0.091-m-) diameter hole in the center of the swirler supplied most of the combustion air. Most of the data were obtained with the 2.9-inch- (0.0736-m-) diameter cones and with the radial swirlers.

In addition to these basic types a number of modifications were tested. They consisted of V-gutter flame spreaders attached to the cans, perforated-metal blockage between the cans at several different axial locations, and mixing tabs and scoops attached

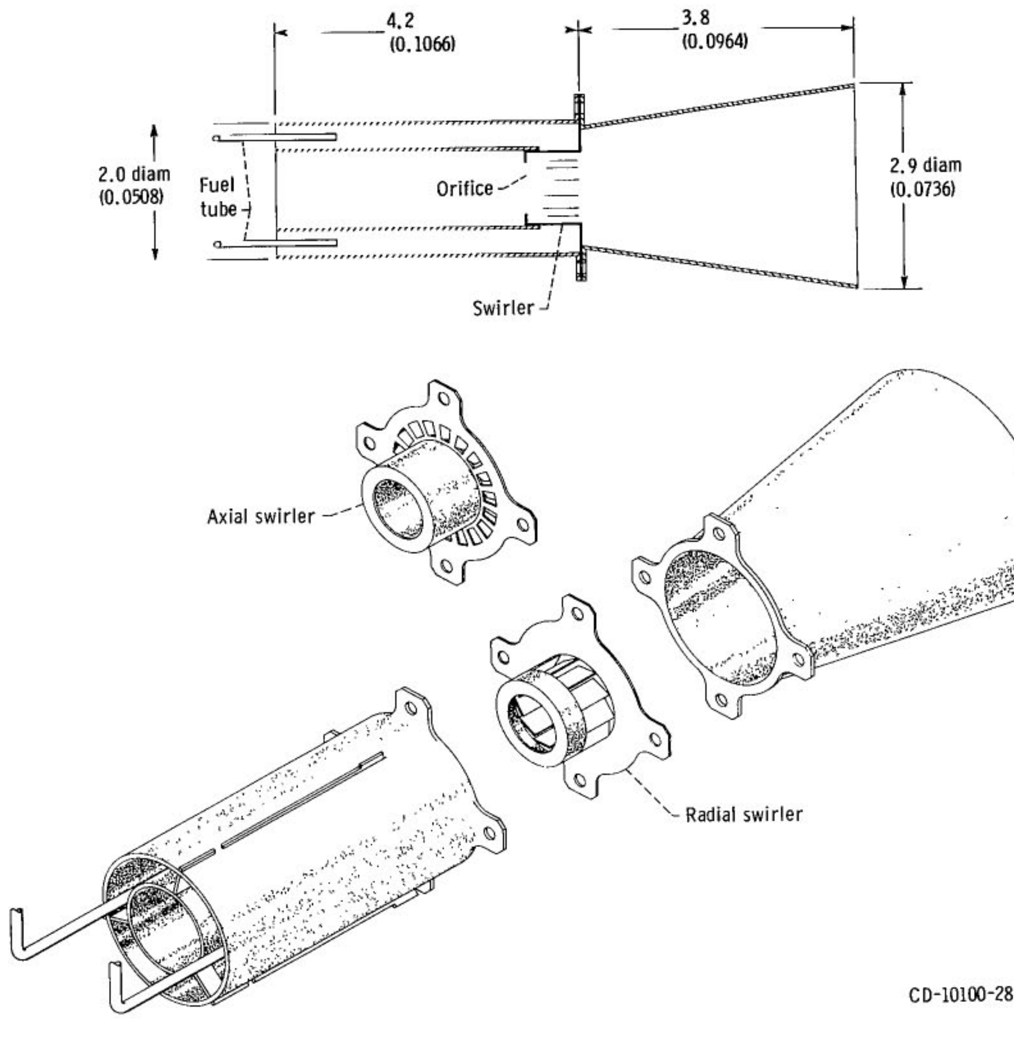

Figure 6. - Swirl-can combustors. (Dimensions are in inches (m).)

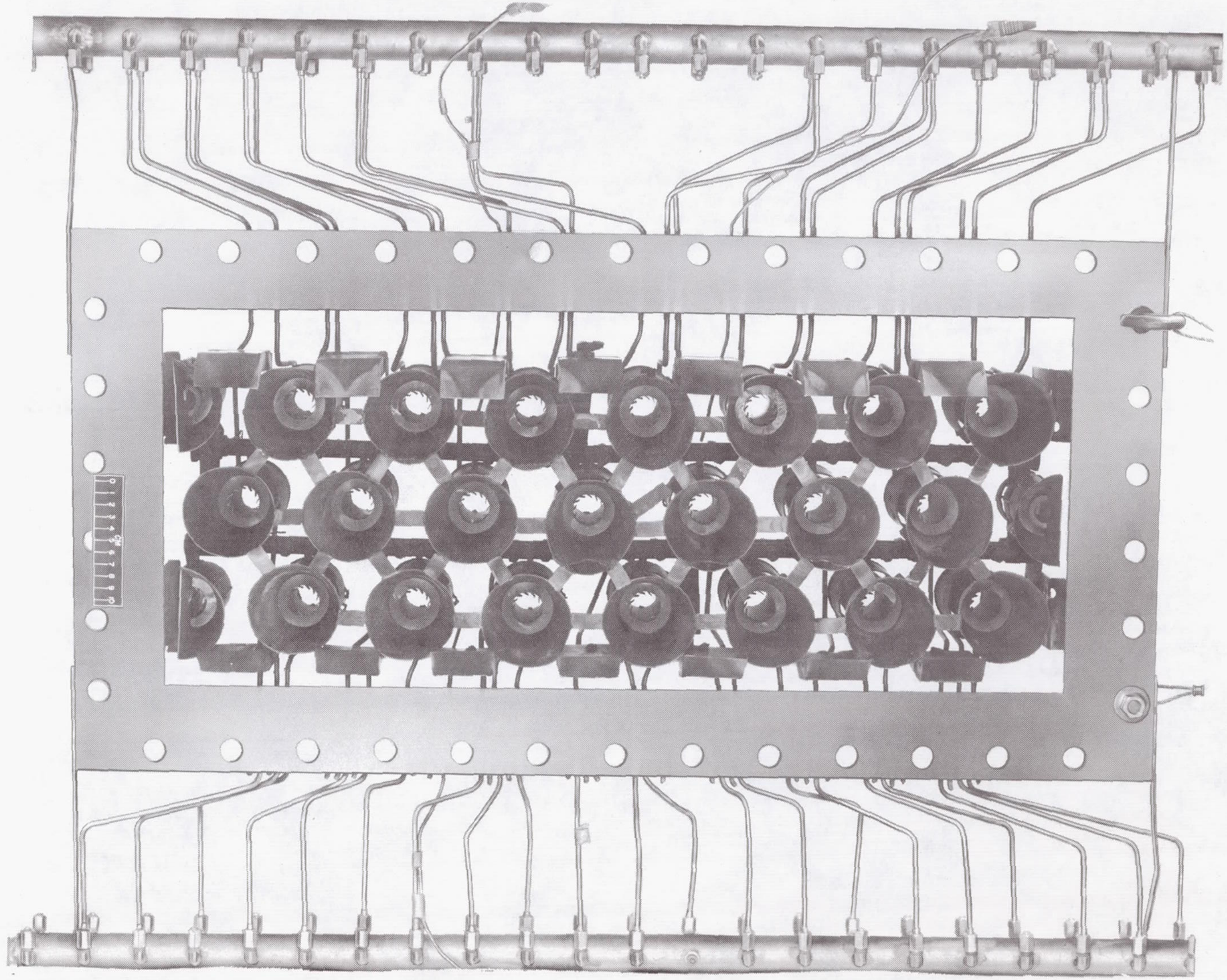

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Figure 7. - Arrangement of swirl-cans in test section.

to the top and bottom cooling liners. Practically all the changes were made in an effort to improve the outlet temperature distribution. Besides the mechanical changes to the swirl-can combustors attempts were made to reduce hot spots in the exhaust profile or to alter the radial profile through adjustments in fuel-flow distribution. A list of the more important modifications tested is presented in table II (p. 13).

A capacitor-discharge-type spark plug with an energy of 20 joules was used to ignite the combustors.

TEST CONDITIONS AND PROCEDURE

Tests were conducted over a range of fuel-air ratios at the combustor-inlet conditions shown in table I:

TABLE I. - COMBUSTOR-INLET CONDITIONS

[Combustor-inlet pressure, 3 atm.]

Test condition	Combustor-inlet temperature		Combustor reference velocity[a]	
	$^{\circ}F$	K	ft/sec	m/sec
1	540	556	119	36.3
2	540	556	150	45.7
3	540	556	190	57.9
4	1140	889	119	36.3
5	1140	889	150	45.7
6	1140	889	190	57.9

[a]Based on maximum cross-sectional area of combustor housing and static pressure and temperature at diffuser inlet.

From a combustion standpoint, the velocity through the combustor is a critical parameter, and hence the term combustor reference velocity is used in judging combustor performance. However, from an engine standpoint, data such as pressure loss and altitude performance can be better correlated if they are expressed in terms of Mach number.

Conditions 1 and 5, except for the pressure, simulate approximately the combustor-inlet conditions that would be encountered at sea-level takeoff and at Mach 3 cruise at an altitude of 65 000 feet (19 812 m), respectively. Average combustor-outlet temperatures of about 2200° F (1478 K) would be required at these conditions. For each test the fuel-air ratio was increased until an average combustor-outlet temperature of 2200° F (1478 K) was attained or until a maximum local temperature of 2700° F (1756 K) was reached.

In addition, a few altitude blowout and reignition tests were made. They were not intended to establish the complete altitude performance of the swirl cans, but were made to get an approximate idea of the altitude reignition capabilities of this type of combustor. Tests were made at a combustor reference Mach number of 0.1 at a fuel-air ratio of 0.01. At a given pressure, the combustor-inlet temperature was reduced gradually until blowout occurred. For the altitude reignition tests, combustor-inlet temperatures were adjusted to a value slightly above that where blowout occurred, and an ignition test was made. If the combustor failed to ignite, the temperature was increased gradually until ignition occurred. Both blowout and altitude reignition tests were made over a range of combustor-inlet pressures.

A jet-type fuel conforming to ASTM A-1 specifications was used throughout the tests. The fuel had an average hydrogen-carbon ratio of 0.161 and a lower heating value of 18 600 Btu per pound (43 300 J/g).

RESULTS AND DISCUSSION

Combustor Development

The various combustor modifications were evaluated according to the following criteria: combustion efficiency, pressure loss, and combustor-outlet temperature distribution. On the whole, the combustion efficiency of all the modifications tested proved to be quite high (85 to 100 percent) and could not be used as a critical parameter to rate the various models. Similarly, pressure loss values did not vary greatly among the various models, except for those modifications where blockage was purposely increased in an attempt to improve mixing. Combustor-outlet temperature distribution, on the other hand, varied appreciably from model to model, and thus became the criterion by which the performance of a particular modification was judged.

A list of the more important combustor modifications investigated is given in table II. Models 1 to 4 are the original designs (two sizes of swirl cans, each with two different swirlers). The other modifications, except for model 14, were made in an attempt to improve mixing and, hence, the temperature profile. Model 14 differed from model 13 only by the addition of narrow strips between cans to promote crossfiring.

The combustor-outlet temperature profiles of models 1 to 4 were rather poor. The 2.9-inch- (0.0736-m-) diameter cans (models 1 and 2) produced profiles with large peaks and deep valleys. The 3.7-inch- (0.094-m-) diameter cans (models 3 and 4) exhibited fairly uniform temperature distribution in the center of the duct, but the temperatures decreased rapidly toward the walls. Hence, it was difficult to achieve high average combustor-outlet temperatures without exceeding the 2700° F (1756 K) local-temperature limit. We believe that the cans were too large to give good coverage of the cross-sectional area without encountering excessive end effects, both at the sides and at the top and bottom. In addition, because of the increased blockage, the pressure loss of

TABLE II. - COMBUSTOR MODIFICATIONS

Model	Maximum swirl-can diameter		Type of swirler	Fuel distribution	Other changes
	in.	m			
1	2.9	0.0736	Radial	Uniform	None
2	2.9	0.0736	Axial	Uniform	None
3	3.7	0.094	Radial	Uniform	None
4	3.7	0.094	Axial	Uniform	None
5	2.9	0.0736	Radial	Uniform	V-gutters attached to cans at downstream edges
6	2.9	0.0736	Radial	Uniform	Perforated-metal blockage between cans in plane of swirlers
7	2.9	0.0736	Radial	Uniform	Perforated-metal blockage between cans at downstream edge of cans
8	2.9	0.0736	Radial	Uniform	Two rows of mixing tabs attached to cooling liner
9	2.9	0.0736	Radial	Reduced fuel to selected cans	Same as for model 8
10	2.9	0.0736	Radial	Top row enriched	Same as for model 8
11	2.9	0.0736	Radial	Top row leaned out	Same as for model 8
12	2.9	0.0736	Radial	Bottom row enriched	Same as for model 8
13	2.9	0.0736	Radial	Uniform	Mixing scoops between cans in top and bottom rows; mixing tabs attached to cooling liner at ramp
14	2.9	0.0736	Radial	Uniform	Same as for model 13 plus narrow cross-fire strips between cans

models 3 and 4 was greater than desired. Tests made with the axial swirlers (models 2 and 4) produced generally poorer profiles (higher peaks and deeper valleys) than those made with the radial swirlers (models 1 and 3). As a result, subsequent tests were restricted to modifications of the 2.9-inch- (0.0736-m-) diameter cans equipped with radial swirlers.

The next attempt at improving the profile consisted of adding V-gutters to the cans (model 5) in order to spread the flames. Only a slight improvement in temperature distribution was realized, not enough to compensate for the increase in pressure loss resulting from the increased blockage.

The next modifications to improve mixing consisted of adding perforated-metal blockage between the cans in two different axial locations: in the plane of the swirlers and in the plane of the can outlets (models 6 and 7). Instead of improving the profile, these changes made the temperature distribution considerably worse. In addition, the pressure drop increased sharply because of the greatly increased blockage. It was concluded that blockage between cans impedes the recirculation of air in the wake of the cans. Hence, this approach was discontinued.

The next modification (model 8) consisted of adding angle tabs to the cooling liner in order to deflect some of the air which would normally sweep along the top and the bottom of the cooling liner toward the center of the can array and thus promote mixing. Two rows of tabs were added, one just downstream of the swirl cans and one at the start of the nozzle ramp. The tabs straddled the cans and protruded about 1 inch (0.0254 m) into the stream. This technique produced a marked improvement in the temperature distribution with a negligible increase in overall pressure loss.

With the geometry of configuration 8 efforts were made to alter the temperature profile, either radial or circumferential, by controlling the fuel flow to the various swirl cans (models 9 to 12). These efforts were only moderately successful. When the temperatures near the side walls were too high, they could be reduced by decreasing the fuel flow to the end cans. Similarly, it was found possible to exercise some control over the radial profile by varying the fuel flow to an entire row of cans. However, where one or two pronounced hot spots were found in the profile, efforts to trace them back to certain swirl cans and then reduce the fuel flow to these cans generally were not successful.

Another attempt at improving mixing was made by replacing the upstream rows of angle tabs with scoops (model 13). The scoops were 2 inches (0.051 m) wide and 0.5 inch (0.013 m) high and were attached to the cans in the top and bottom rows so as to straddle the cans (figs. 2 and 7). The discharge opening of the scoops was 2 inches (0.051 m) by 1.4 inches (0.036 m). This modification generally produced the best results, although the improvement over model 8 was not great.

The data presented in this report (table III), with the exception of the altitude blowout and reignition tests, are those obtained with model 13. Blowout and reignition tests

TABLE III. - COMBUSTOR PERFORMANCE DATA

(a) Combustion tests. Model 13; pressure, 3 atmospheres.

Run	Inlet-air temperature		Airflow		Nominal reference velocity		Fuel-air ratio	Average combustor-outlet temperature		Combustion efficiency, percent	Combustor pressure loss, $\Delta P/P$, percent	Temperature distribution parameters		Temperature rise variation ratio, ΔTVR
	^{o}F	K	lb/sec	kg/sec	ft/sec	m/sec		^{o}F	K			∂_{stator}	∂_{rotor}	
1	547	559	37.50	17.01	119	36.3	------	----	---	-----	3.55	-----	-----	----
2	540	556	36.68	16.64			0.0099	1161	901	91.78	3.91	0.407	0.115	1.40
3	544	558	36.99	16.78			.0147	1486	1081	95.98	4.24	.402	.127	1.44
4	545	558	36.85	16.71			.0196	1805	1258	99.24	4.25	.342	.122	1.34
5	542	556	37.03	16.80			.0217	1943	1335	100.75	4.39	.333	.114	1.36
6	541	556	46.07	20.90	150	45.7	0.0098	1157	898	91.11	6.02	0.439	0.120	1.40
7	544	558	45.90	20.82			.0149	1481	1078	94.23	6.46	.441	.128	1.42
8	544	558	45.99	20.86			.0196	1789	1250	97.89	6.58	.379	.125	1.35
9	544	558	46.05	20.89			.0235	2023	1379	98.97	7.13	.318	.115	1.38
10	537	554	58.43	26.50	190	57.9	.0097	1092	862	83.18	9.71	.481	.127	1.54
11	541	556	58.56	26.56	190	57.9	0.0097	1108	871	85.09	9.84	0.467	0.110	1.47
12	542	556	58.38	26.48			.0148	1415	1042	88.58	10.41	.491	.125	1.47
13	541	556	58.40	26.49			.0197	1758	1232	95.39	11.19	.413	.124	1.44
14	542	557	58.42	26.50			.0215	1855	1286	95.30	11.49	.402	.120	1.43
15	543	557	58.27	26.43			.0234	1946	1337	94.43	11.04	.469	.104	1.47
16	1103	868	22.30	10.12	119	36.3	-----	----	----	------	2.37	----	----	----
17	1119	877	28.21	12.80	150	45.7	-----	----	----	------	2.53	----	----	----
18	1123	879	35.59	16.14	190	57.9	-----	----	----	------	5.42	----	----	----
19	1120	878	22.26	10.10	119	36.3	0.0100	1770	1239	102.38	2.31	0.359	0.085	1.30
20	1108	871	22.45	10.18	119	36.3	.0127	1935	1330	103.92	3.52	.319	.086	1.37
21	1109	871	22.21	10.07	119	36.3	0.0160	2115	1431	101.89	2.68	0.375	0.083	1.44
22	1129	883	28.14	12.76	150	45.7	.0089	1683	1190	97.16	3.68	.293	.061	1.23
23	1141	889	28.46	12.91			.0123	1919	1322	101.02	3.75	.248	.069	1.26
24	1120	877	28.54	12.95			.0159	2126	1437	102.69	3.94	.219	.069	1.22
25	1123	882	28.08	12.74			.0181	2234	1496	100.67	4.02	.211	.062	1.24
26	1130	883	28.10	12.75	150	45.7	0.0069	1550	1117	93.75	3.62	0.453	0.016	1.36
27	1136	887	36.08	16.37	190	57.9	.1007	1759	1233	97.88	6.03	.299	.070	1.36
28	1129	883	36.06	16.36			.0128	1929	1327	99.89	6.30	.288	.068	1.31
29	1136	887	35.90	16.28			.0155	2084	1413	99.75	6.24	.300	.062	1.37
30	1133	885	36.26	16.45			.0168	2169	1460	100.87	6.40	.302	.068	1.36

TABLE III. - Concluded. COMBUSTOR PERFORMANCE DATA

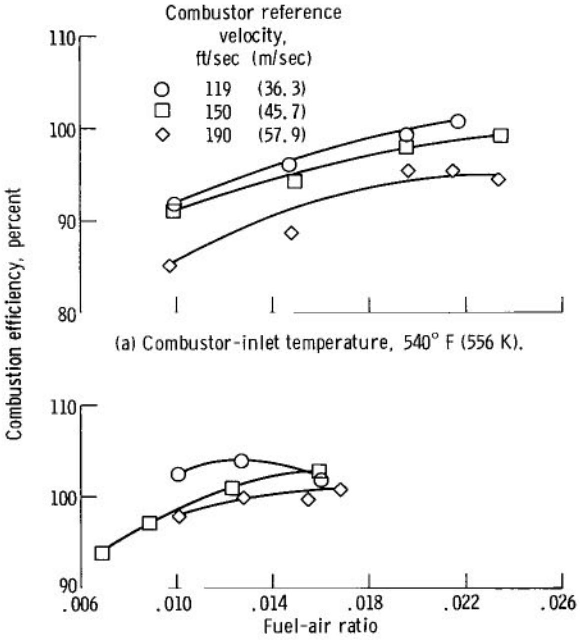
(b) Blowout and reignition tests. Model 14; reference Mach number, 0.1; fuel-air ratio, 0.01.

Run	Inlet-air pressure, atm (absolute)	Inlet-air temperature		Air flow		Ignition	Blowout
		^{o}F	K	lb/sec	kg/sec		
1	1.36	400	478	22.56	10.23	Yes	No
2	↓	350	450	23.25	10.55	↓	↓
3	↓	300	422	24.00	10.89	↓	↓
4	↓	250	394	24.83	11.26	↓	↓
5	↓	200	367	25.76	11.68	↓	↓
6	1.36	150	339	26.79	12.15	Yes	No
7	1.36	100	311	27.97	12.69	↓	↓
8	1.16	400	478	19.18	8.70	↓	↓
9	1.16	300	422	20.40	9.25	↓	↓
10	1.16	250	394	21.10	9.57	↓	↓
11	1.16	200	367	21.89	9.93	Yes	No
12	1.16	150	339	22.77	10.33	↓	↓
13	1.16	100	311	23.77	10.78	↓	↓
14	1.02	400	478	16.92	7.67	↓	↓
15	1.02	350	450	17.44	7.91	↓	↓
16	1.02	300	422	18.00	8.16	Yes	No
17	1.02	250	394	18.62	8.45	Yes	↓
18	1.02	200	367	19.32	8.76	No	↓
19	.88	350	450	15.11	6.85	Yes	↓
20	.88	300	422	15.60	7.08	No	↓
21	0.75	450	506	12.06	5.47	Yes	No
22	↓	400	478	12.41	5.63	No	↓
23	↓	350	450	12.79	5.80	↓	↓
24	↓	300	422	13.20	5.99	↓	↓
25	↓	250	394	13.66	6.20	↓	Yes
26	.61	450	506	9.87	4.48	↓	Yes

were made with model 14. Preliminary tests suggested that the addition of narrow strips between cans improved crossfiring during ignition but did not significantly affect the other combustion parameters.

Combustion Efficiency

Combustion efficiency data obtained with model 13 at three reference velocities and at two inlet-air temperatures are presented in figure 8. Combustion efficiency increased with increasing inlet-air temperature and with decreasing reference velocity. At the maximum fuel-air ratios that could be attained without exceeding local-temperature limits of 2700° F (1756 K), combustion efficiencies generally ranged between 95 and 100 percent. The combustion efficiency obtained at the 1140° F (889 K) combustor-inlet temperature condition at times exceeded 100 percent. It is believed that improper sampling is primarily responsible. At the end of the tests made at this condition, it was noticed that the top thermocouple had been bent slightly out of position. In general, the data indicate that combustion efficiency should be no problem with this type of combustor.



(b) Combustor-inlet temperature, 1140° F (889 K).

Figure 8. - Effect of combustor-inlet conditions on combustion efficiency of model 13. Combustor-inlet pressure, 3 atmospheres.

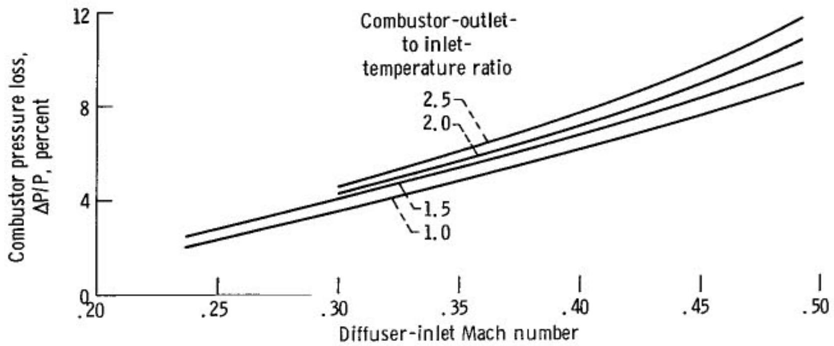


Figure 9. - Effect of diffuser-inlet Mach number on pressure loss of model 13. Combustor-inlet pressure, 3 atmospheres.

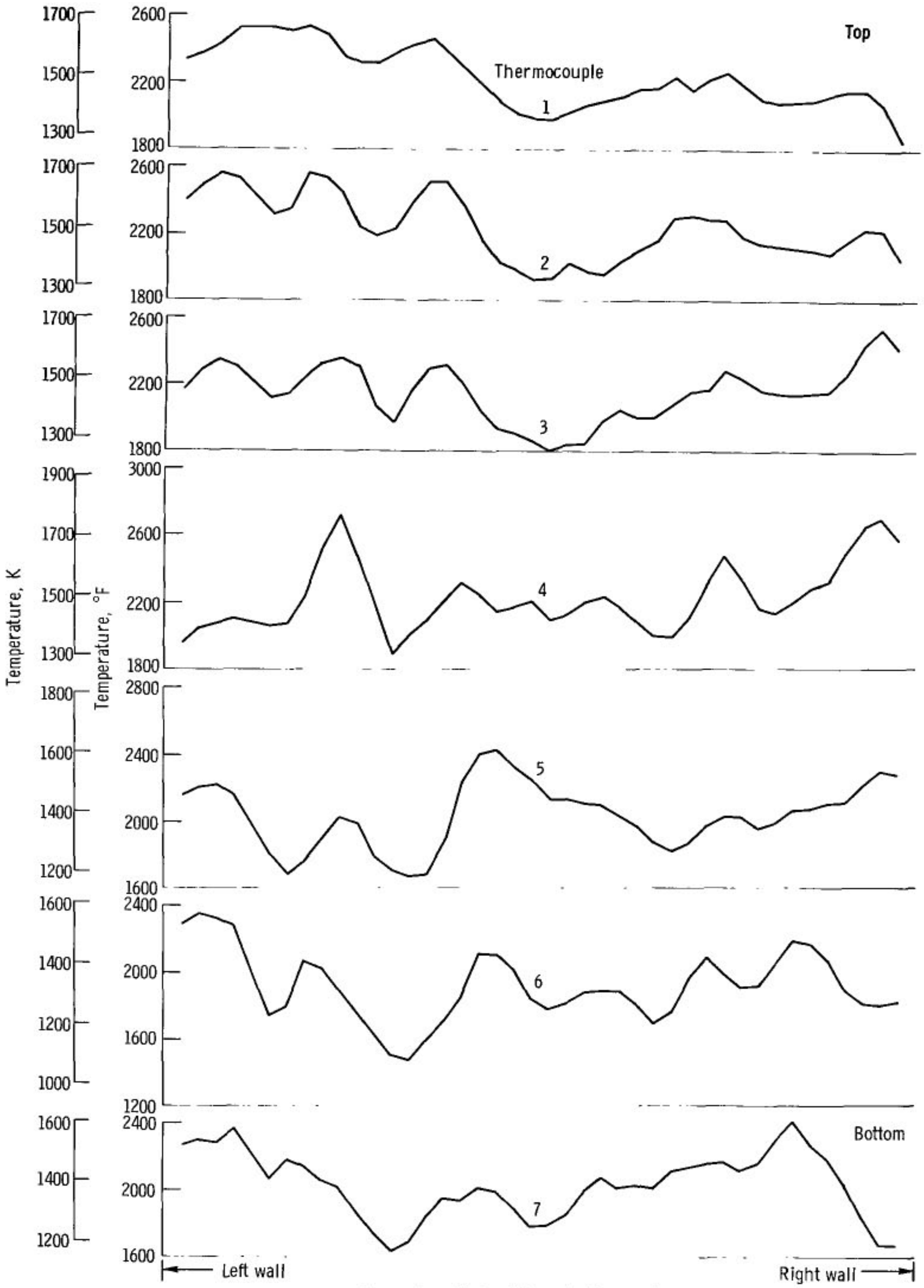
Pressure Loss

In figure 9 values of pressure loss $\Delta P/P$ are plotted against diffuser-inlet Mach number for four combustor-outlet- to inlet-temperature ratios. Pressure loss increased with increasing Mach number and with increasing temperature ratio. At a diffuser-inlet Mach number of 0.3 and a combustor-outlet- to inlet-temperature ratio of 2.5, the pressure loss, including the diffuser pressure drop, was about 4.6 percent.

Temperature Distribution

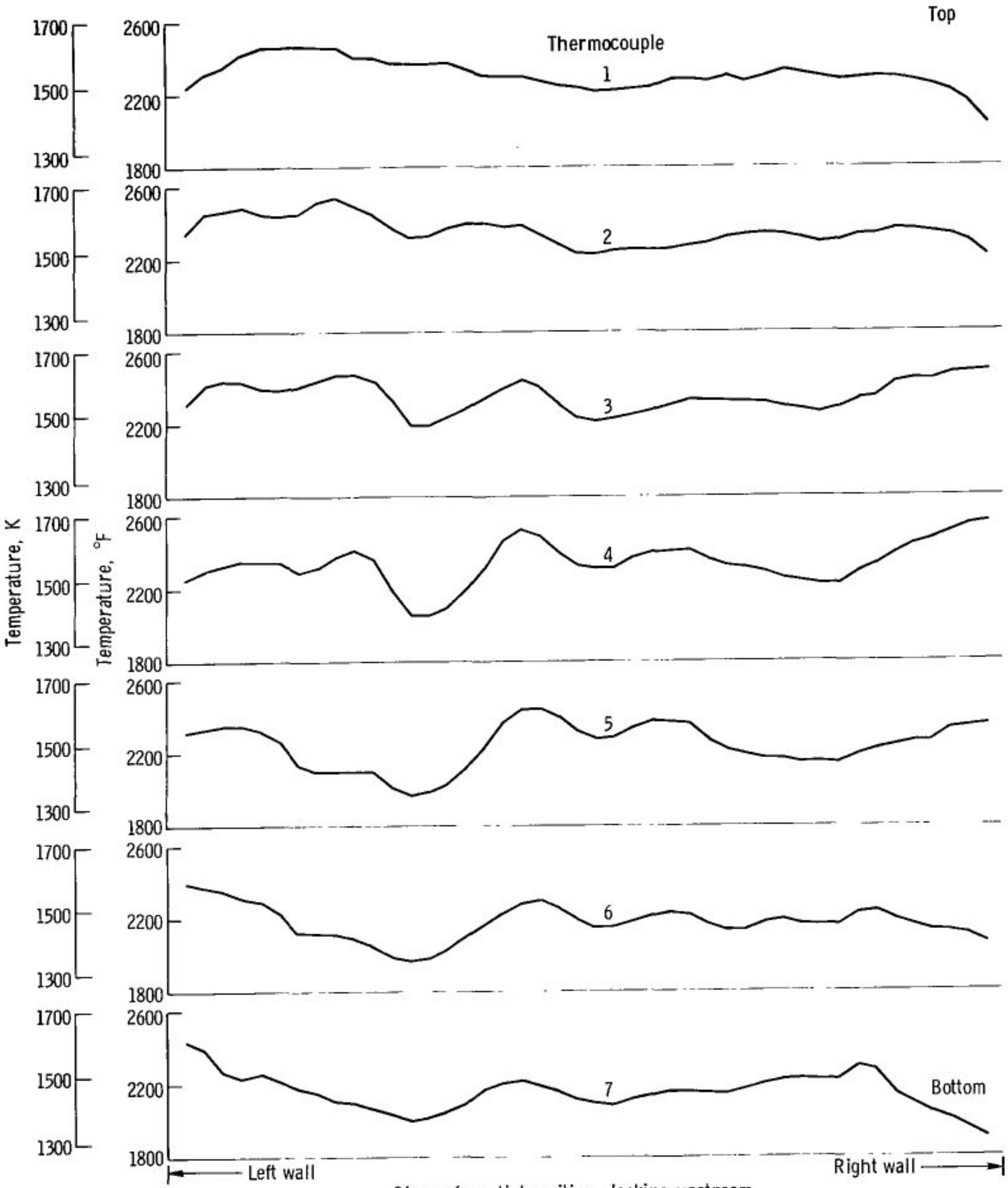
Circumferential temperature profiles, for each of the seven positions on the exhaust rake, are presented in figure 10. At a reference velocity of 150 feet per second (45.7 m/sec), a combustor-inlet temperature of 540° F (556 K), and a fuel-air ratio of 0.0235, maximum temperature differences of approximately 850° F (472 K) were observed between the highest and lowest temperatures in any radial position. The maximum temperature at any position was 2720° F (1766 K), and the average temperature, excluding 10 percent at each side, was 2116° F (1431 K). The values of δ_{stator} and δ_{rotor} for this condition were 0.318 and 0.115, respectively.

At the same reference velocity, but at a combustor-inlet temperature of 1140° F (889 K) and a fuel-air ratio of 0.0181, the temperature distribution of this model (fig. 10(b)) was considerably improved. The maximum temperature difference between the highest and lowest temperature at any one radial position was approximately 520° F (289 K). The maximum temperature was 2545° F (1670 K) with an average of 2269° F (1516 K). The values of δ_{stator} and δ_{rotor} at this condition were 0.211 and 0.062, respectively.



(a) Inlet-air temperature, 540° F (556 K); corrected average combustor-outlet temperature, 2116° F (1431 K).

Figure 10. - Temperature distribution of model 13. Inlet-air pressure, 3 atmospheres; combustor reference velocity, 150 feet per second (45.7 m/sec).

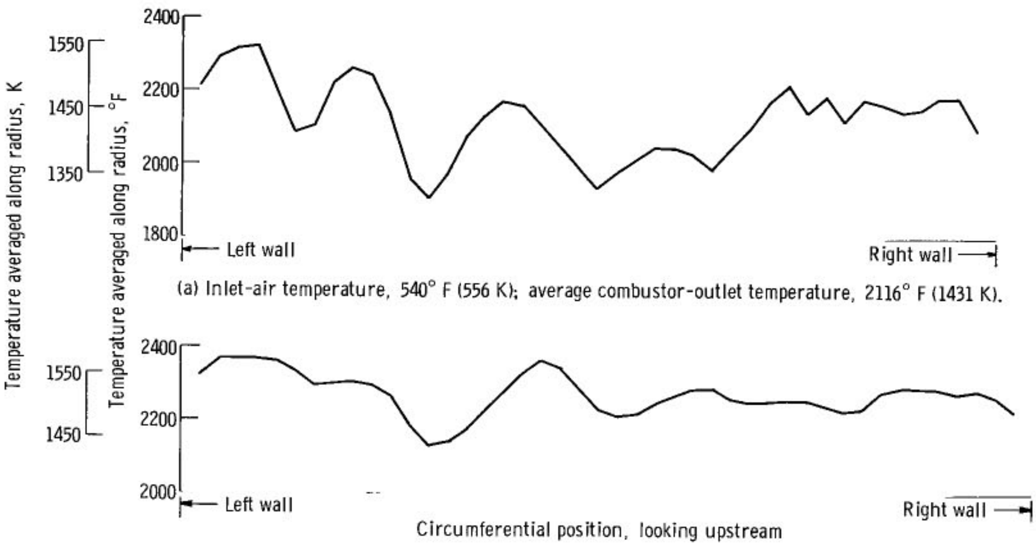


(b) Inlet-air temperature, 1140° F (889 K); corrected average combustor-outlet temperature, 2269° F (1516 K).

Figure 10. - Concluded.

The values of ΔTVR for these two conditions were 1.38 and 1.24, respectively. Thus, the temperature distribution improved noticeably with increasing combustor-inlet temperature. Also, from an inspection of the profiles (fig. 10) at both temperature conditions, it is apparent that the major problem is one of local hot spots. Velocity profile surveys made in the diffuser suggest that considerable improvement in combustor-outlet temperature distribution could be made by improving the circumferential velocity profile at the combustor inlet.

The combustor-outlet temperatures, averaged along a radius and plotted against circumferential position, are shown in figure 11 for the same conditions. Again, it is evident that the profile improved appreciably as the combustor-inlet temperature was increased from 540° to 1140° F (556 to 889 K).



(b) Inlet-air temperature, 1140° F (889 K); average combustor-outlet temperature, 2269° F (1516 K).

Figure 11. - Average circumferential combustor-outlet temperature profile of model 13. Inlet-air total pressure, 3 atmospheres; combustor reference velocity, 150 feet per second (45.7 m/sec).

Average radial temperature profiles for model 13 at the same combustor-inlet conditions are shown in figure 12. Here, at each of seven radial positions, combustor-outlet temperatures are averaged circumferentially, and the difference between these values and the average temperature for all seven radial positions is plotted against radial position, expressed as percentage of combustor-outlet height. The ideal radial profiles shown on these plots are representative of the requirements of current supersonic turbojet engines. At a combustor-inlet temperature of 540° F (556 K) a maximum deviation from the average of 200° F (111 K) was observed, while at 1140° F (889 K) the maximum deviation was 120° F (67 K). The average radial profile matched the ideal

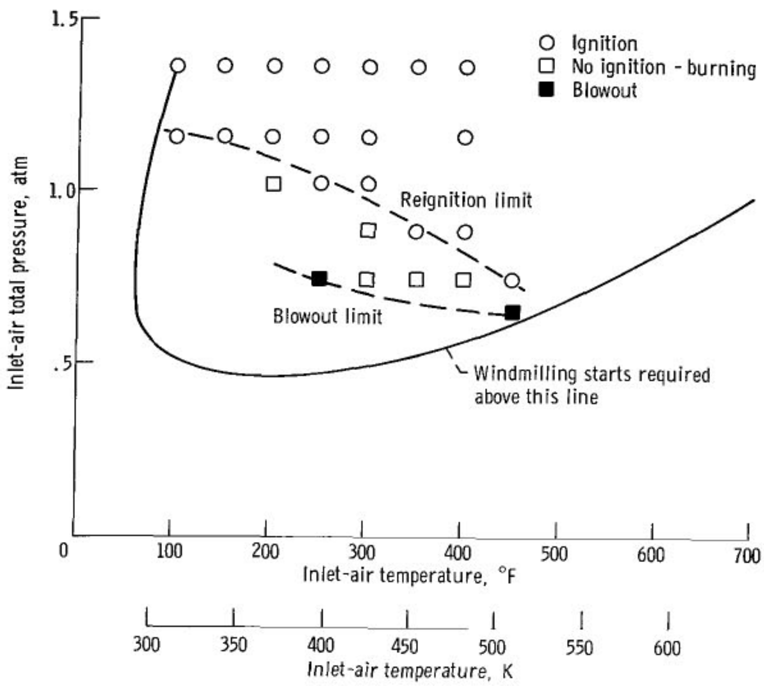


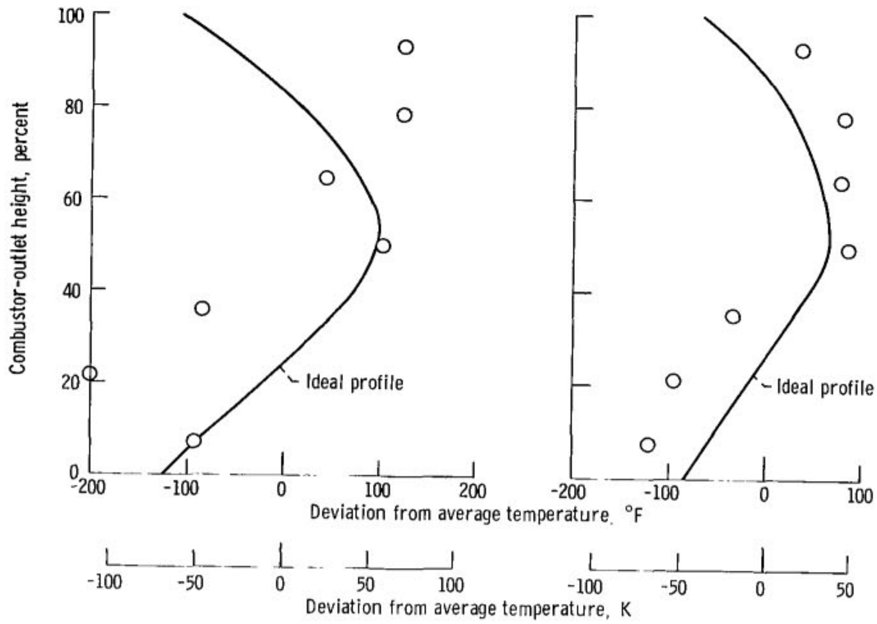
Figure 13. - Reignition and blowout tests of model 14. Combustor reference Mach number, 0.1; fuel-air ratio, 0.01.

profile more closely at the high-combustor-inlet-temperature condition than at the low-temperature condition. All of the data with model 13 were obtained with uniform fuel distribution; as previously mentioned, some success in adjusting the radial profile through control of fuel flow was demonstrated with other models.

Blowout and Reignition

Altitude blowout and reignition performance data obtained with model 14 are shown in figure 13. As previously mentioned, these tests were not intended to map out the altitude performance of this model but were included solely to get an approximate idea of the altitude capabilities of this type of combustor. No efforts to improve altitude reignition, such as increasing spark energy or relocating the spark plug, were made. Similarly, no changes were made to the swirl cans to improve stability.

Tests were conducted at a combustor reference Mach number of 0.1 and at a fuel-air ratio of 0.01. At a given pressure the combustor-inlet temperature was decreased



(a) Inlet-air temperature, 540° F (556 K); corrected average combustor-outlet temperature, 2116° F (1431 K).

(b) Inlet-air temperature, 1140° F (889 K); corrected average combustor-outlet temperature, 2269° F (1516 K).

Figure 12. - Average radial profile of model 13. Inlet-air total pressure, 3 atmospheres; combustor reference velocity, 150 feet per second (45.7 m/sec).

until blowout occurred. After blowout the combustor-inlet temperature was increased in small increments until ignition took place.

The data in figure 13 show that at a combustor-inlet pressure of 1.16 atmospheres no blowout occurred at inlet temperatures as low as 100° F (311 K); reignition was possible at all temperatures. As the pressure was reduced to 0.75 atmosphere, blowout occurred at an inlet temperature of 250° F (394 K). For reignition at this pressure the temperature had to be raised to 450° F (505 K). At 0.61 atmosphere blowout occurred at an inlet-air temperature of 450° F (505 K). The reignition limits are affected by the particular ignition system used. The blowout limits represent the reignition limits which might be attained with a properly designed ignition system.

In general, the swirl cans performed well at the reduced inlet-air pressure and temperature conditions. The flames were blue, and no streaking was observed. Crossfiring between cans was good; at all conditions investigated, all the cans were ignited.

The solid curve shown in figure 13 outlines an area of pressure and temperature conditions encountered in a typical advanced supersonic aircraft. Above this boundary line windmilling starts are required. At the low inlet-air pressures and temperatures, the swirl-can combustors in their present design would not meet this requirement.

SUMMARY OF RESULTS

A number of swirl-can combustor modifications were tested with ASTM A-1 fuel in a 21-can-array test section at the following conditions: combustor length (can outlet to exhaust nozzle), 20 inches (0.508 m); combustor-inlet pressure, 3 atmospheres; combustor-inlet temperatures, 540° and 1140° F (556 and 889 K); reference velocities, 119, 150, and 190 feet per second (36.3, 45.7, and 57.9 m/sec).

The best modification produced the following results:

1. In the range of fuel-air ratios for an average combustor-outlet temperature of 2200° F (1478 K), combustion efficiencies near 100 percent were obtained.

2. At a diffuser-inlet Mach number of 0.3 and a combustor-outlet- to inlet-temperature ratio of 2.5, the overall pressure loss (including diffuser) $\Delta P/P$ was 4.6 percent.

3. The combustor-outlet temperature distribution improved appreciably with increasing combustor-inlet temperature. At a reference velocity of 150 feet per second (45.7 m/sec), an inlet temperature of 540° F (556 K), and a fuel-air ratio of 0.0235, the temperature distribution parameters δ_{stator} and δ_{rotor} had values of 0.318 and 0.115, respectively. At the same reference velocity, but at an inlet temperature of 1140° F (889 K) and a fuel-air ratio of 0.0181, the values of these parameters were 0.211 and 0.062, respectively.

4. At a combustor reference Mach number of 0.1, a fuel-air ratio of 0.01, and a pressure of 1.16 atmospheres, no blowout occurred at inlet temperatures as low as 100° F (311 K). At a pressure of 0.75 atmosphere, blowout occurred at an inlet-air temperature of 250° F (394 K). For reignition at this condition, it was necessary to increase the inlet-air temperature to 450° F (505 K).

CONCLUDING REMARKS

The results of this investigation are sufficiently encouraging to suggest that the concept of combustors made up of individual swirl-can elements could be extended to full-scale combustors. The data show that combustion efficiency and pressure loss requirements should present no serious problems. The combustor-outlet temperature distribution presented some problems, but there was sufficient improvement with increasing inlet-air temperature to produce acceptable temperature profiles. Furthermore, we believe that in future applications substantial improvements in combustor-outlet temperature distribution could be made through changes in airflow distribution at the combustor inlet. At the same time, additional improvements in temperature distribution might be brought about by changes in the size and arrangement of the individual combustor elements.

The greatest shortcoming of the swirl-can combustors was their altitude reignition capability. Additional development is required to improve the low-pressure, low-temperature ignition limits and combustion stability limits.

Lewis Research Center,
National Aeronautics and Space Administration,
Cleveland, Ohio, October 8, 1968,
126-15-02-50-22.

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